ATV/ISS NAVIGATION STRATEGIES WITH GPS

Jose Miguel LOZANO-GONZALEZ* – Juan Antonio BÉJAR-ROMERO - Francisco M. MARTINEZ-FADRIQUE*****

GMV S.A., c/ Isaac Newton 11, P.T.M.-Tres Cantos, E-28760 Madrid, SPAIN
**Tel : +34 91 807 21 29 / Fax : 21 99 / email:jmlozano@gmv.es*
***Tel : +34 91 807 33 22 / Fax : 21 99 / email:jabejar@gmv.es*
****Tel : +34 91 807 21 38 / Fax : 21 99 / email:fmartinez@gmv.es*

Jean François GOESTER* – Pascal DESMAZEAUX**

Centre National d'Etudes Spatiales, 18 avenue Edouard Belin, 31400 Toulouse, France
**Tel : +33 05 61 16 78 / Fax : 05 61 27 35 40 / email:jean-francois.goester@cnes.fr*
***Tel : +33 05 61 28 33 45 / Fax : 05 61 27 35 40 / email: pascal.desmazeaux@cnes.fr*

ABSTRACT *- This paper presents an analysis of the absolute and relative navigation strategies, using GPS measurements, in the frame of ATV mission. The analysis covers the absolute navigation of ATV during phasing and de-orbiting, and the relative navigation between ATV and ISS during rendezvous phase, using State Vector Differences and Single Differences algorithms and EKF and LSM filters. In addition, the influence of operational constraints, such as visibility, sampling frequency and data gaps, in navigation performances is analyzed.*

KEYWORDS: ATV, ISS, GPS, Navigation, Extended Kalman Filter, Least Squares Method, Orbit Determination.

INTRODUCTION

CNES is responsible for developing the Flight Dynamics Subsystem of the Automated Transfer Vehicle Control Center that will perform tasks such as ground maneuver calculations, orbit determination and onboard Guidance Navigation and Control monitoring. The Figure 1 represents the different phases of a typical ATV mission.

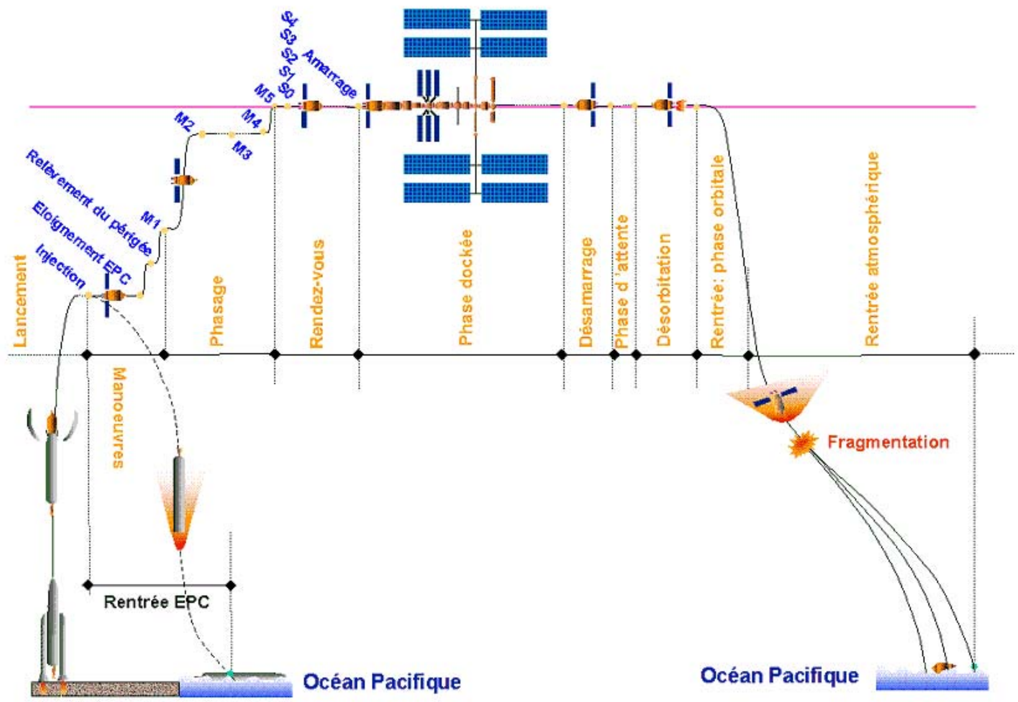

Figure 1 Different phases of ATV mission

One of the most critical Flight Dynamics issues is the real-time navigation of the ATV.

During phasing and de-orbiting, the nominal absolute navigation (localization of ATV vehicle) is performed on ground.

During rendezvous phase in accordance with the ISS crew security constraints, the relative GPS filter is different from the onboard one and allows cross-checking process to detect some abnormal ATV navigation behavior. Both real-time and off-line modes can be activated using different algorithms to perform comparison. The off-line mode is based on the Least Square Method (LSM), whereas the real-time mode is based on an Extended Kalman Filter (EKF). The real-time navigation filter is run as soon as GPS measurements are downloaded to ATV-CC via TM and gives an orbit solution. The off-line navigation processes a set of GPS measurements. It is activated before any optimization of maneuvers to:

- Take into account GPS measurements stored onboard during visibility holes, and compare the result with the Kalman Filter state vector.

- Update the nominal trajectory as needed.

- Calibrate Orbital Control System thruster level during a burn.

This paper presents the strategies implemented for the navigation performed on ground, and the expected performance in real-time operations.

OPERATIONAL CONSTRAINTS

The following subsections introduce the main constraints related to ATV GPS-based Orbit Determination.

Real Time Data Availability

Due to ATV real-time navigation needs, the possibility to used GPS precise ephemeris can not be considered. Instead of this type of ephemeris, the information contained in the navigation message file is used to know the GPS satellite ephemeris.

The effects of this operational constraint are analyzed in the simulation results section.

Data gaps

During the development of a typical ATV mission, if any problem appears in the ATV GPS receiver or in the data downloaded via TM, the Orbit Determination behavior must be analyzed.

The effects of gaps in the measurements have been taken into account and are shown in the simulation results section.

MODELIZATION

In the frame of this study GMV have developed a prototype of a GPS-based Orbit Determination Tool, consisting of a GPS measurement tool and a GPS orbit determination tool, suitable to ATV needs. The following sections summarize the main issues related to GPS-based orbit determination applicable to ATV and therefore to the objectives of this study.

In this section a description of the models used in the development of this study is made.

GPS Constellation

To obtain the GPS measurement is necessary to know in the measurement epoch the spacecraft ephemeris and the GPS satellite ephemeris. The following lines introduce the algorithm to generate the GPS satellite ephemeris.

Due to operational constraints, the baseline for the ATV-CC is the navigation message information.

In order to analyze the impact of the use of the navigation message and not precise ephemeris, two possibilities to determine the GPS satellite ephemeris are considered. The first one using precise ephemeris files, the SP3 files (Standard Product #3) and the second one considering the information contained in the navigation message files.

SP3 files considered are the NIMA precise ephemeris files. The SP3 file format is described in [1]. These precise ephemeris are not available in real-time, so they are considered only to compare the results obtained with the navigation message.

The algorithm to obtain the GPS satellite ephemeris for a certain epoch using the information available in these precise ephemeris files is the following one:

- Search, in the ephemerides data for this satellite, the epochs t_i and t_{i+1} that fulfil the following equation:

$$t_i \leq \text{epoch} < t_{i+1}$$

- Perform a Lagrange Interpolation using non-equal spaced points, taking n/2 points in the left side of t_i (including t_i) and n/2 points in the right side of t_{i+1} (including t_{i+1}). So the number n of points to be used in the interpolation must be an even number. This interpolation is used to generate the position and velocity of the GPS satellite.

In order to have real-time access to the satellite positions and satellite system time, the orbit information contained in the navigation message is used. The parameters contained in the navigation message are used to compute the GPS satellite time and coordinates considering the dynamical model taken from [2].

Ionospheric model

The ionospheric perturbation due to the GPS signal travel through the ionosphere can not be calculated from the measurements because only one frequency is available.

In order to correct the measurements, it was decided to use the integration of electron density along the ray path with IRI-95 electron density model [3].

Orbit Determination Dynamical Model

The following lines make a brief description of the models considered in the orbit determination process.

Spacecraft's Dynamical Model

The following models have been considered:

- GEM10B gravity model.

- CIRA88-MSIS86 atmospheric model. Cannonball aerodynamics

- Third-bodies perturbation is not considered.

State Transition Matrix

The variational equation formulation is not performed in the traditional way of integration of the variational partials. The variational partials are substituted by the computation of the state transition matrix by means of simplified analytical model.

The extended state vector, position, velocity and additional parameters; in each time, can be expressed using the extended state vector in the initial time:

$$X(t) = \phi(t, t_k) X(t_k)$$

where ϕ is the state transition matrix.

To obtain this state transition matrix it is used a simplified model based on the following hypothesis:

- The orbit of the spacecraft is a keplerian one.
- The partial derivatives contained in the state transition matrix are obtained by analytical differentiation of the keplerian solutions.

Pseudo-range measurement model and error sources

The GPS observable considered in the ATV/ISS navigation strategies are the pseudo-ranges from code measurements. In this section, the algorithm to generate these measurements is described.

The following equation represents the algorithm used to simulate the GPS pseudo-range measurements in the simulation tool:

$$PR = |\vec{x}_{GA}| - \vec{x}_{GA}\frac{\vec{v}_G}{c} + E_{ion} + E_{apccog} + c\Delta T_{AR} + c\Delta T_{RC} - c\Delta T_{GC} + c\Delta T_{REL} + c\Delta T_{SA} + E_{MP} + E_T$$

The following equation represents the algorithm used to simulate the GPS pseudo-range measurements in the orbit determination tool:

$$PR = |\vec{x}_{GA}| - \vec{x}_{GA}\frac{\vec{v}_G}{c} + E_{ion} + E_{apccog} + c\Delta T_{AR} + c\Delta T_{RC} - c\Delta T_{GC} + c\Delta T_{REL}$$

which can be divided in:

$\lvert \vec{x}_{GA} \rvert - \vec{x}_{GA} \dfrac{\vec{v}_G}{c}$	$\Rightarrow$	geometrical measurement (m)
E_{ion}	$\Rightarrow$	ionospheric refraction correction (m)
E_{apccog}	$\Rightarrow$	GPS antenna phase center to COG correction (m)
ΔT_{AR}	$\Rightarrow$	antenna-receiver delay (s)
ΔT_{RC}	$\Rightarrow$	receiver clock error (s)
ΔT_{GC}	$\Rightarrow$	GPS satellite clock error (s)
ΔT_{REL}	$\Rightarrow$	relativistic correction (s)
ΔT_{SA}	$\Rightarrow$	selective availability (s) considered only in simulation tool
E_{MP}	$\Rightarrow$	multipath error (m) considered only in simulation tool
E_T	$\Rightarrow$	random error (m) considered only in simulation tool

where $\vec{x}_{GA}$ is the relative position vector, i.e. from the spacecraft to the GPS Satellite, and $\vec{v}_G$ is the GPS velocity vector.

In the following lines the different error sources and its simulation is described.

Ionospheric Refraction Correction

The ionospheric refraction correction is due to the signal crossing through the ionosphere, and the subsequent change in the refraction coefficient with respect to the refraction coefficient in the vacuum. This correction is implemented with the following model:

$$E_{ion} = \frac{40.3 \cdot TEC}{f_{GPS}^2}$$

where f_{GPS} is the carrier frequency of the GPS signal and TEC is the Total Electron Count. The Total Electron Count is the number of electron that the signal encounters during its travel in the ionosphere. The algorithm to obtain the TEC is the following: integration along the signal path in the ionosphere is performed,

$$TEC = \int_{path} EC(r)$$

where EC(**r**) is the number of electron in a certain position of the ionosphere (**r**). The value of EC is obtained from the model IRI-95. This model is described in [3].

GPS Antenna Phase Center Correction

The fundamental GPS observable is the signal travel time between the GPS satellite antenna and the spacecraft receiver antenna. The signal travel time is scaled into range measurement using the signal propagation velocity.

When the position of a GPS satellite is known, it is known the position for the COG, Center Of Gravity. The GPS signal leaves the GPS satellite at the antenna phase center not the center of gravity.

The GPS antenna phase center correction takes into account the difference between the range measured from the center of gravity (COG) and the range from the antenna phase center.

Antenna-Receiver Delay

Physically, the measurement is made from antenna to antenna, but there is a delay due to the fact that the measurement is evaluated in the receiver, so there is a delay between the reception of the signal in the antenna and the processing of the signal in the receiver. This error term is modeled like a constant term error.

Receiver Clock Error

The time scale used for GPS constellation is the GPS time. Both the receiver and the GPS satellite clock must represent this time but they have some errors.

In the case of the receiver clock in the simulation tool, this error is modeled with a linear error term plus a random walk term:

$$\Delta T_{RC} = E_0 + E_1(t - t_0) + Random\_Walk(\sigma_{RC})$$

The coefficients of the linear error terms are tool inputs. The random walk term is implemented in the following way:

$$Random\_Walk(t_0) = 0$$
$$Random\_Walk(t_i) = Random\_Walk(t_{i-1}) + N(0, \sigma_{RC})$$

where $N(0, \sigma_{RC})$ is a value of a gaussian law with mean value equal to zero and with a standard deviation equal to σ_{RC}, which is a tool input.

When the measurement is generated in the orbit determination tool, this error is modeled with a second-order polynomial:

$$\Delta T_{RC} = E_0 + E_1(t - t_0) + E_2(t - t_0)^2$$

The coefficients of the error terms are taken from the inputs of the tool, in the case in which they are not going to be estimated, or from the last estimation, in the case in which they are going to be estimated.

GPS Satellite Clock Error

The clocks of the GPS satellites are not perfect, and they have a certain error with respect to the real GPS time. This clock error is modeled depending on the GPS model used to know the position of the GPS satellite.

If precise ephemeris files are used, the following equation models the clock error:

$$\Delta T_{GC} = a_0 + a_1(t - t_0) + Random\_Walk(\sigma_{GC})$$

The linear error terms are generated using the values of the clock error available in the precise ephemeris file. A linear interpolation is performed between the two nearest clock errors to the simulation epoch.

If navigation message files are used, then the model is represented by:

$$\Delta T_{GC} = a_0 + a_1(t - t_0) + a_2(t - t_0)^2 + Random\_Walk(\sigma_{GC})$$

The coefficients of the above equation are available in the navigation message files. A linear interpolation is performed between the two nearest clock errors to the simulation epoch.

The random walk term is implemented in the following way:

$$Random\_Walk(t_0) = 0$$
$$Random\_Walk(t_i) = Random\_Walk(t_{i-1}) + N(0, \sigma_{GC})$$

where $N(0, \sigma_{GC})$ is a value of a gaussian law with mean value equal to zero and with a standard deviation equal to σ_{GC}, which is a tool input.

The random walk term is considered only in the simulation tool. It is also interpolated like the linear error terms between the two nearest random walk contributions.

Relativistic Correction

The effect of the theory of the relativity in the measurement simulation is double. There is a contribution due to the space-time curvature, that is negligible for the required accuracy, and there is another contribution due to each of the clock, because the clocks are in movement and this movement introduces a time gap.

The error term due to each clock is modeled in the following way:

$$\Delta T_{SC} = -4.443 \cdot 10^{-10} e\sqrt{a}\sin(E)$$

where e is the eccentricity of the orbit, a is the semi-major axis and E is the eccentric anomaly in the measurement epoch. This correction must be applied to ΔT_{RC} and ΔT_{GC}, so in the relativistic correction the effect of spacecraft clock is positive and the effect of GPS satellite clock is negative due to the negative sign of ΔT_{GC} in the measurement model. The relativistic correction is, taking into account the three terms, is:

$$\Delta T_{REL} = \Delta T_{SC} - \Delta T_{GPS}$$

Selective Availability

The model used for the selective availability is the following one:

$$\left.\begin{array}{l} \left.\begin{array}{l} n_0 = \mathrm{Int}\left(\dfrac{T}{2dt}\right) \\ \sigma_{SA} \end{array}\right\} \Rightarrow \left.\begin{array}{l} X(t_i) = N(0, \sigma_{SA}) \\ t_i = t - 2n_0 dt, \ldots, t \end{array}\right\} \Rightarrow \\ W_0 = 0.42 + 0.5\cos(\pi x) + 0.08\cos(2\pi x) \end{array}\right\} \Rightarrow y(k) = \sum_{i=-2n_0}^{i=0} W_i X(t_i)$$

A characteristic time (T) and a standard deviation σ_{SA} define this model. The model is based in the generation of a set of gaussian values for a certain number of epochs (n_0) previous to the present one, and the ponderation of these values with a weighting function (W). From one epoch to the next one, there is a shift in the set of gaussian values, the older one is discarded and a new gaussian value is generated for the present epoch.

This error is only considered in the simulation tool.

Multipath Error

The effect of the multipath can not be modeled in the frame of this model, because its effect is not an increment in the measurement but a problem for the receiver hardware to correlate correctly the GPS received signal with the one generated by the receiver. Despite this, a multipath error term has been added to the model to taking into account a random error due to this effect. This error term is for each measurement a value of a gausssian law with a different standard deviation for each measurement.

This error is only considered in the simulation tool.

Random Error

This error term is used to consider all the random effects that have not been considered explicitly in the measurement model. This error term is for each measurement a value of a gausssian law with a different standard deviation for each measurement. This standard deviation value is for each measurement the precision that can be obtained with each type of measurement. Usually, these values are around 1 meter for pseudo-range.

This error is only considered in the simulation tool.

NAVIGATION ALGORITHM

Roughly speaking the GPS-based Orbit Determination or Navigation System is in charge of computing the chaser/target absolute and relative position and velocity. Processing observables derived from the GPS signal measurements coming from the chaser and target GPS receivers performs this. The major characteristic elements, for every concept, are:

- GPS observable to be processed.
- Algorithm in charge of obtaining the optimum estimate of the chaser-target relative position and velocity by processing the above mentioned observable together with a relative orbit prediction algorithm, when required.

The GPS observables to be processed are built from the GPS signal measurements, which are supplied by the GPS receiver. This has been briefly described earlier.

The following orbit determination systems, based on processing different observables, have been considered for the development of the ATV GPS-based Orbit Determination Tool.

Absolute Navigation

The objective of the absolute navigation is to obtain the absolute position, velocity, and related variables (such as thrust level, drag effect, GPS receiver clock estimation,…) of the chaser (ATV), while minimizing the impact of the different error sources (ionosphere effects, GPS satellites clock error,…). The navigation performance is based on the trade-off between accuracy and frequency of processed measurements (convergence delay, and precision).

Relative Navigation

Two different strategies have been considered in the relative navigation

Relative navigation with absolute state vector differences

This relative navigation concept is the simplest one and it consists in computing the target/chaser (ATV/ISS) relative state vectors from the difference of the target and chaser absolute state vectors. Each absolute state vector is obtained by means of the LSM and EKF algorithms. This method implements a quite robust relative navigation concept where each spacecraft process its own absolute position and velocity in an independent way. This kind of navigation is in charge of combining the information coming from the orbit prediction algorithm and from the GPS measurements. The GPS satellites to be tracked by the chaser and the target GPS receivers can be different. This reduces drastically the constraints imposed by the GPS antennae pointing direction in the definition of the mission reference profiles (orbit and attitude) and allows every GPS receivers to optimize its own set of visible and selected GPS satellites.

Relative navigation with single differences

In this case, the relative navigation concept is based on an estimation of the target/chaser relative state vector by processing, whatever the filtering algorithm, the differences between the GPS measurements coming from the chaser and the target GPS receivers (ATV and ISS).

The difference is performed between measurements corresponding to the same GPS satellites. The use of the single differences produces the cancellation of some errors common to both GPS receiver measurements and at least the reduction of the effects of some of them:

- Errors associated to the GPS satellite are cancelled (errors due to the GPS satellite ephemeris, GPS satellite clock).
- The effect of the GPS signal propagation errors, such as the signal delay introduced by the ionosphere, is reduced due to the spatial proximity of target and chaser.

The estimated state vector, in this case, is the same than in the relative navigation with absolute state vector differences. The difference between both algorithms is the type of measurement processed by the filter, raw measurement in one case and difference of raw measurements in the last case. This imposes the necessity of using simultaneous measurements from both spacecraft.

Anyway, during the rendezvous phase, the chaser and the target spacecraft are very close each other and the GPS antennae pointing configuration are the same, so the set of selected GPS is usually the same for both spacecraft.

ORBIT DETERMINATION ALGORITHMS

The basis of the orbit determination algorithm is to obtain a new estimation of the extended state vector (position, velocity and additional parameters), using the last estimation and the models defined for the dynamics of the spacecraft and for the measurements. From the last extended state vector, the same measurements available from the GPS receiver are simulated, and the difference between the real and simulated measurement is used to feed an estimation filter to obtain the new extended state vector.

For this study a Bayesian Least Squares method (LSM) has been used off-line mode and an Extended Kalman Filter (EKF) for the real-time navigation.

SIMULATION RESULTS

In this section the results of the absolute and relative navigation strategies are shown.

Visibility Constraint

The number of visible GPS satellites depends on the antenna gain pattern.

The antenna gain pattern considered in the results of this study consists of a conical surface defined by the semi-angle value.

In the following figure, Figure 2, it is shown the number of visible satellites for the considered antenna gain pattern.

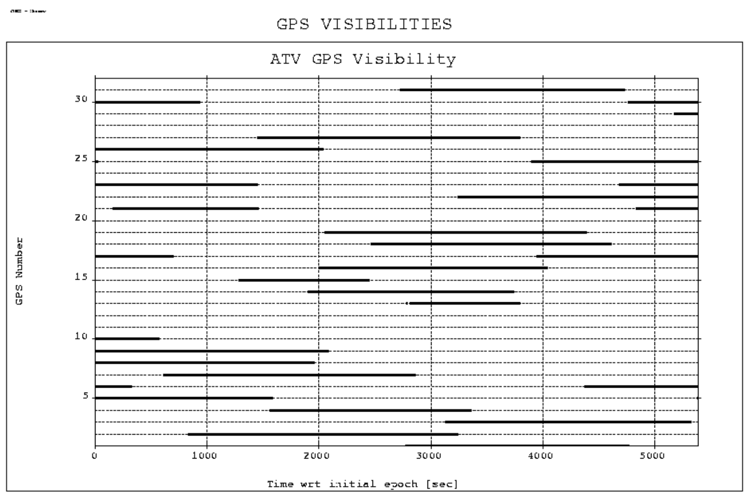

Figure 2 GPS Visibilities with Semi-angle Value set to 85º

Absolute navigation

The following lines show the results of the implemented solution to faced up to the operational constraints.

Real Time Data Availability

Due to real time constraints, the precise ephemeris for GPS satellites cannot be considered as a solution to know the GPS ephemeris. In order to compare the orbit determination results with GPS precise ephemeris and the ones obtained considering the navigation message information, this case is considered.

In this test case the filter used is EKF.

In the following two figures, Figure 3 and Figure 4, the impact of using navigation message instead of precise ephemeris is shown.

These figures show the difference in ATV estimated position module between the navigation message and the precise ephemeris. Selective Availability is considered in Figure 4.

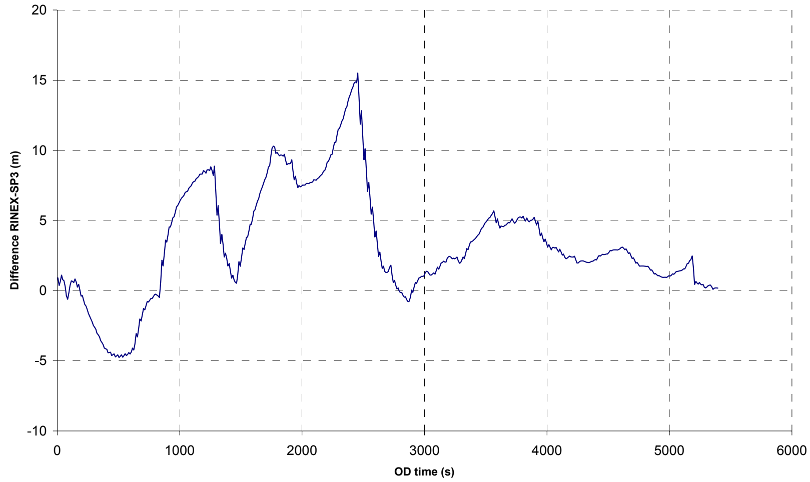

Figure 3 Difference in ATV Estimated Position Module (SA off)

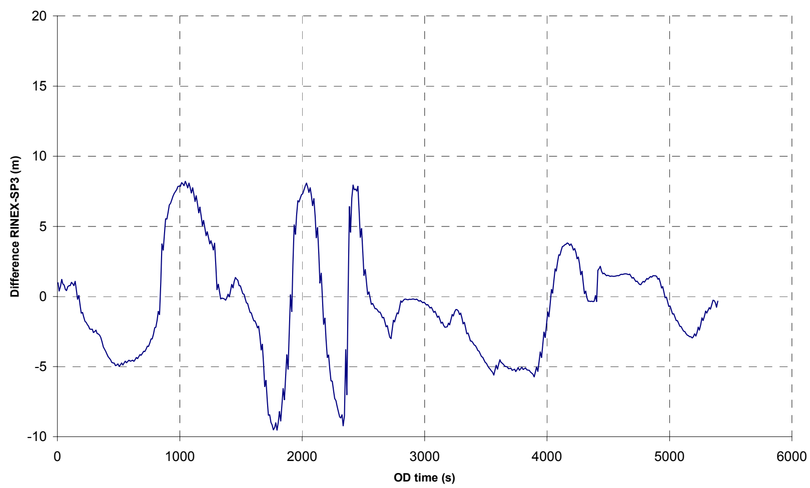

Figure 4 Difference in ATV Estimated Position Module (SA on)

The following two figures represent the estimated position using navigation message information and the associated residuals when the SA is not considered.

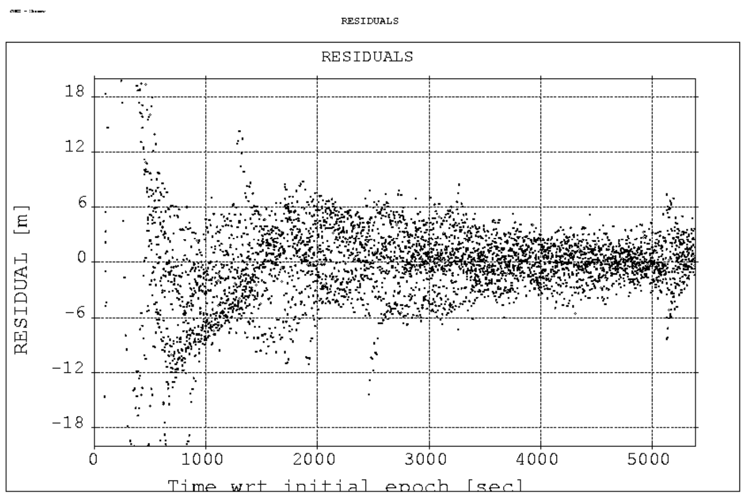

Figure 5 Residuals using Navigation Message

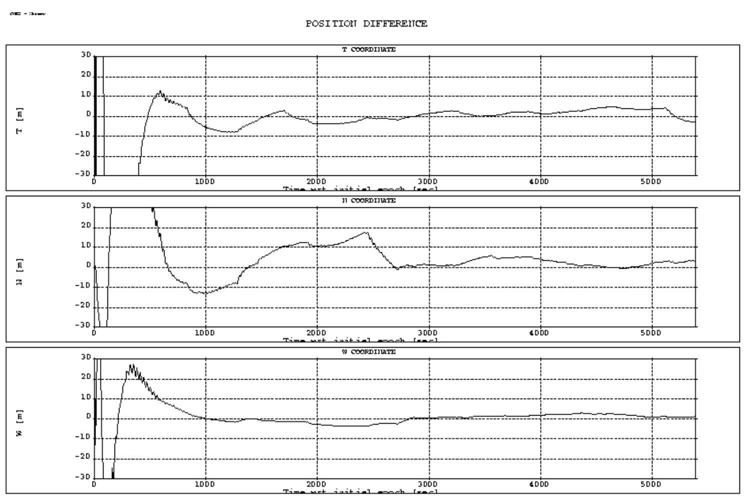

Figure 6 Estimated Position Error using Navigation Message

The following two figures represent the estimated position using navigation message information and the associated residuals when the SA is considered.

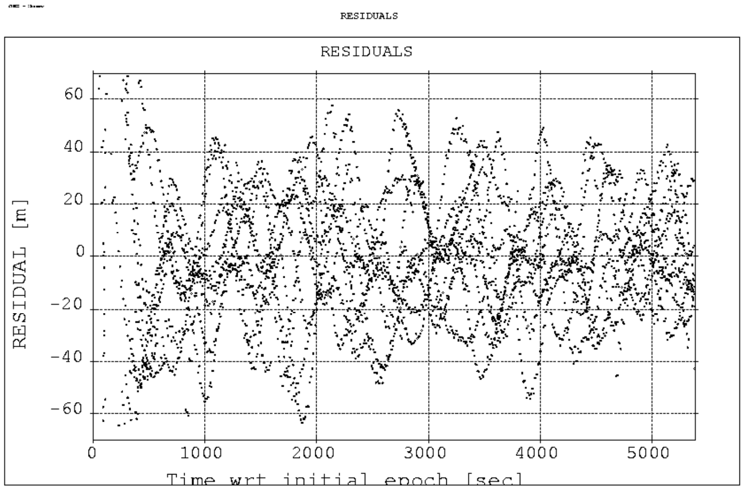

Figure 7 Residuals using Navigation Message and with SA

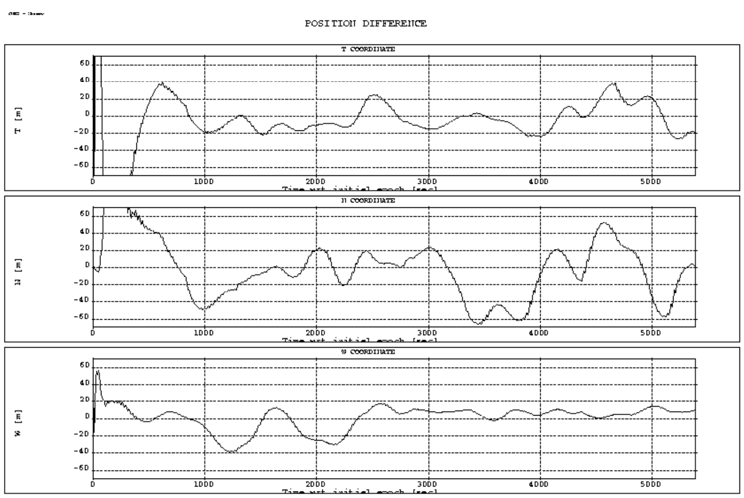

Figure 8 Estimated Position Error using Navigation Message and with SA

The following table represents the difference between the estimated trajectories in terms of percentage of orbit determination time in which the accuracy in position module is below a certain value. In this case the SA is not considered.

Table 1 Percentage of Time depending on the Position Error (SA off)

	Percentage of time below 10 m	Percentage of time below 20 m	Percentage of time below 30 m	Percentage of time below 40 m	Percentage of time below 50 m
Precise Ephemeris	85 %	88 %	90 %	90 %	91 %
Navigation Message	62 %	89 %	90 %	91 %	91 %

In the following table same information is shown when the SA is considered:

Table 2 Percentage of Time depending on the Position Error with SA

	Percentage of time below 10 m	Percentage of time below 20 m	Percentage of time below 30 m	Percentage of time below 40 m	Percentage of time below 50 m
Precise Ephemeris	3 %	17 %	49 %	61 %	73 %
Navigation Message	3 %	19 %	46 %	61 %	74 %

The differences in position accuracy and orbit determination behavior using precise ephemeris or navigation message are compliant with the accuracy requirement (50 m in position, 3σ) and allow the real-time navigation using GPS pseudo-ranges measurements.

Measurement gaps

The behavior of the orbit determination process when there is a measurement gap is analyzed in this subsection.

The duration of the gap is parameterized in order to compare the evolution of the orbit determination. It is also compared the behavior of the batch algorithm and the real-time algorithm.

The following figure, Figure 9, shows the difference in the value of the RMS of the position module error depending on the duration of the data gap and on the filter used.

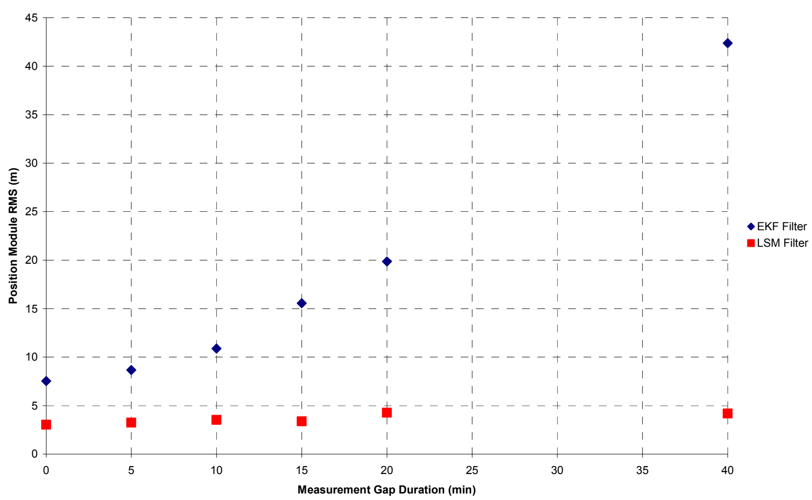

Figure 9 RMS Value of Position Module Error

The following figure, Figure 10, shows the difference in the value of the RMS of the velocity module error depending on the duration of the data gap and on the filter used.

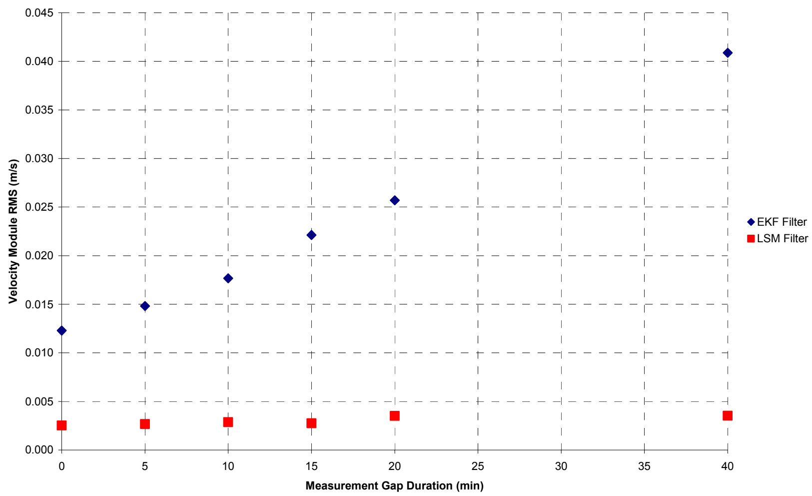

Figure 10 RMS Value of Velocity Module Error

To analyze the recovery performances in the real-time navigation, the time in which the required accuracy in position, 50 m (3σ), is not reached after the measurement gap will be defined as the recovery time.

The following table shows the recovery performance analysis results.

Table 3 Recovery Performances Results in Absolute Real-Time Navigation

Gap Duration (min)	Recovery Time (min)	K
5	< 0.5	0
10	< 0.5	0
15	< 0.5	3
20	2.5	5
40	24.5	6

Where K represents the number of rejected measurements from the first available set of measurements after the measurement gap.

The time span considered for the previous analysis is one and a half hour, that is, the LSM filter process in each iteration a set of measurements covering one and a half hour, while the EKF filter covers the same time span with filtering steps of 30 seconds.

Relative navigation

Sampling frequency

In this subsection the influence of the number of measurements available in each filter run is analyzed.

The sampling frequencies of the measurements that are considered in these test cases are 1 Hz, 0.1 Hz, and 0.01 Hz.

The following table shows the results of the sampling frequency.

Table 4 Sampling Frequency Analysis Results using State Vector Differences

Sampling Frequency (Hz)	Convergence Delay (s)	Percentage of time below 1 m	Percentage of time below 2 m	Percentage of time below 5 m	Percentage of time below 10 m
0.01	305	10 %	40 %	86 %	96 %
0.1	325	65 %	93 %	96 %	97 %
1	< 60	98 %	99 %	99 %	99 %

Table 5 Sampling Frequency Analysis Results using Single Differences

Sampling Frequency (Hz)	Convergence Delay (s)	Percentage of time below 1 m	Percentage of time below 2 m	Percentage of time below 5 m	Percentage of time below 10 m
0.01	100	6 %	36 %	82 %	99 %
0.1	< 60	66 %	95 %	99 %	99 %
1	< 60	99 %	99 %	99 %	99 %

Convergence delay is defined as the period of time elapsed until the module position accuracy, 10 m (3 σ), is reached.

The percentage of time corresponds to the percentage of the orbit determination time in which the module position error is smaller than each mentioned position accuracy.

The nominal sampling frequency selected is 0.1 Hz, one set of measurements every 10 seconds. This frequency represents a trade-off solution between the accuracy and the convergence behavior.

Real Time Data Availability

In this section the comparison between using precise ephemeris files for the GPS satellites and the information contained in the navigation message is done. Considering real time constraints, the precise ephemeris for GPS satellites can not be considered as a solution to know the GPS ephemeris. Due to the relative navigation algorithms used, the differences between the estimated orbits should be smaller than in the absolute navigation.

In this test case the filter used is EKF.

The following figure, Figure 11, shows the difference between the estimated position module error when the state vector differences algorithm is used.

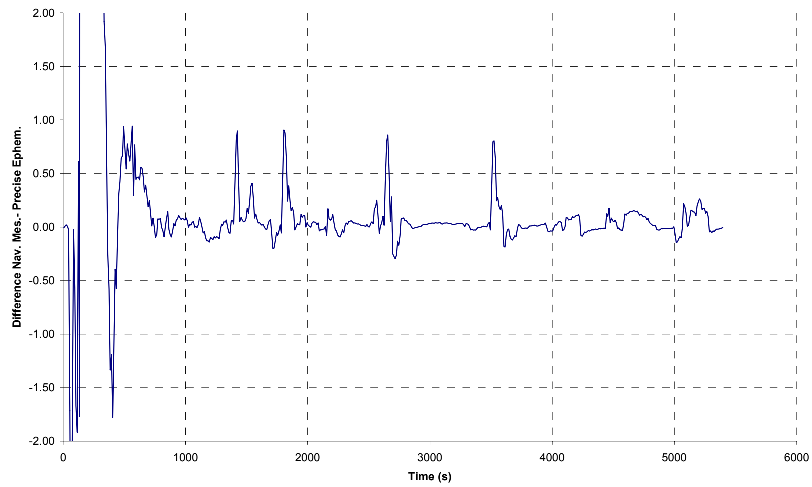

Figure 11 Difference in Position Module Error using State Vector Differences

In the following figure, Figure 12, the single differences algorithm is used.

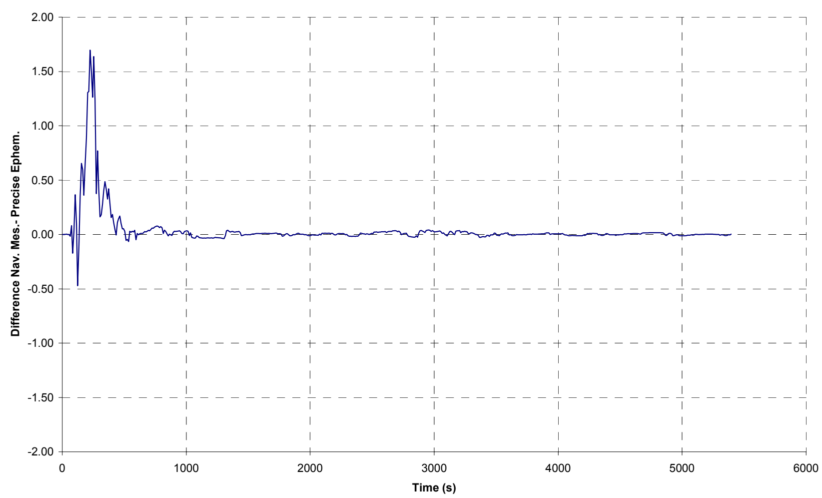

Figure 12 Difference in Position Module Error using single Differences

The figures show how the influence of the GPS ephemeris precision is reduced when the state vector differences algorithm is used and it can be considered that there is not influence in the case of single differences algorithm.

The following figures, Figure 13 and Figure 14, represent the results in orbit determination when the state vector differences algorithm is used.

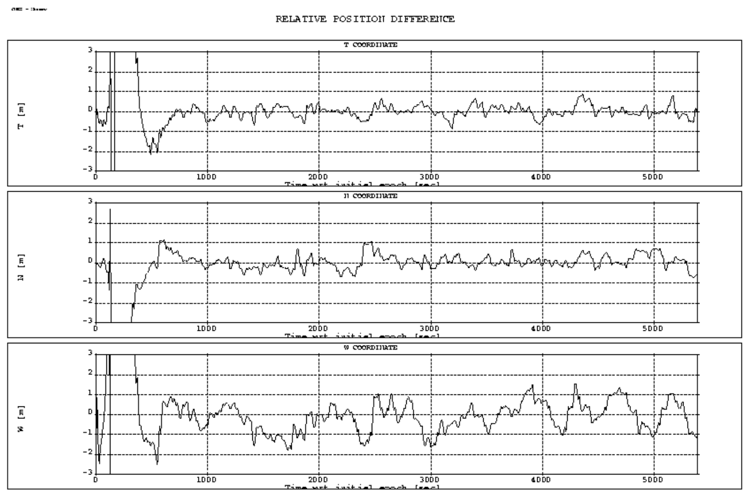

Figure 13 Position Error using State Vector Differences and Navigation Message

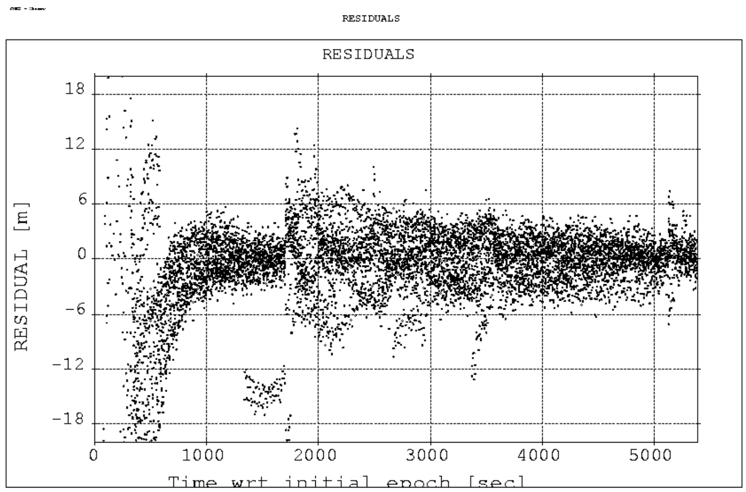

Figure 14 Residuals using State Vector Differences and Navigation Message

The following figures, Figure 15 and Figure 16, represent the results in orbit determination when the single difference algorithm is used.

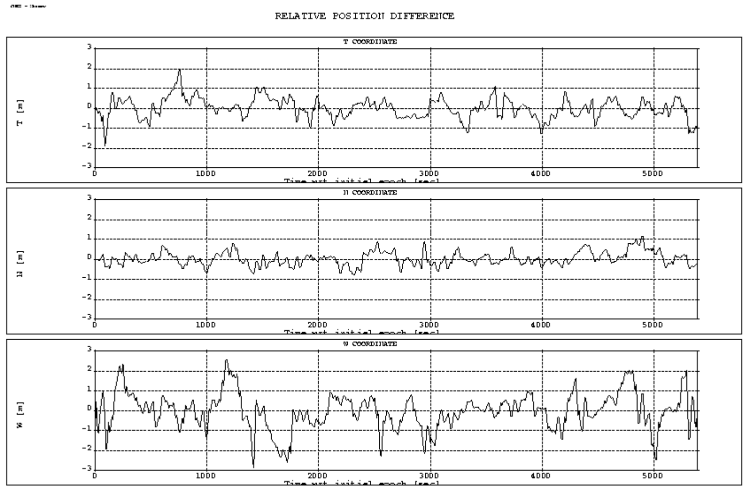

Figure 15 Position Error using Single Differences and Navigation Message

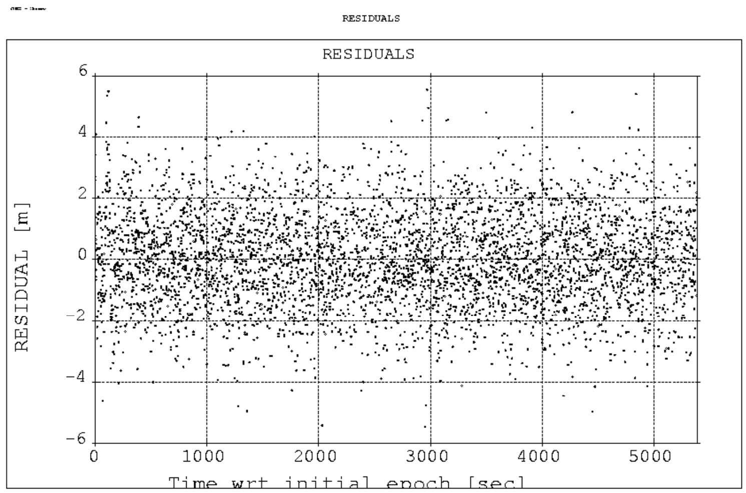

Figure 16 Residuals using Single Differences and Navigation Message

SA Influence

When the Single Difference algorithm is used, the errors common to both GPS receivers are cancelled, so the SA influence is cancelled if this algorithm is used.

If the State Vector difference is used and with the particular configuration of the ATV mission, SA errors in both SC is similar so when the difference is done this influence is also reduced.

The following graphs, Figure 17 and Figure 18, show the difference in position module error if the SA is considered or not.

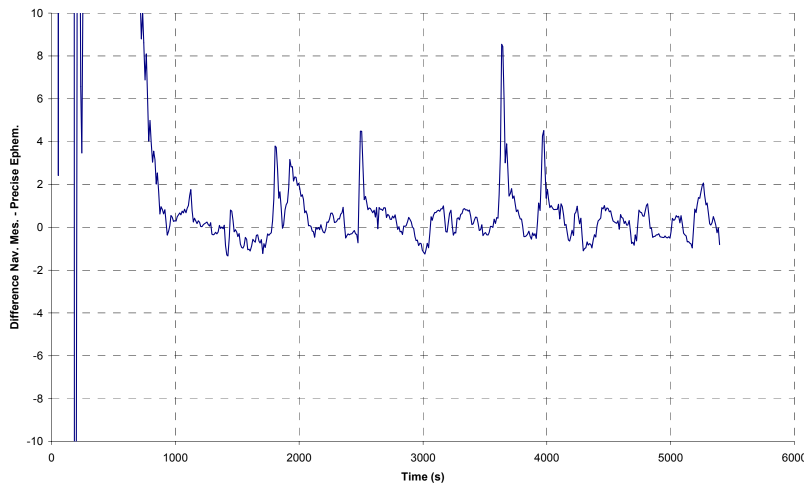

Figure 17 Difference in Position Module Error using State Vector Differences

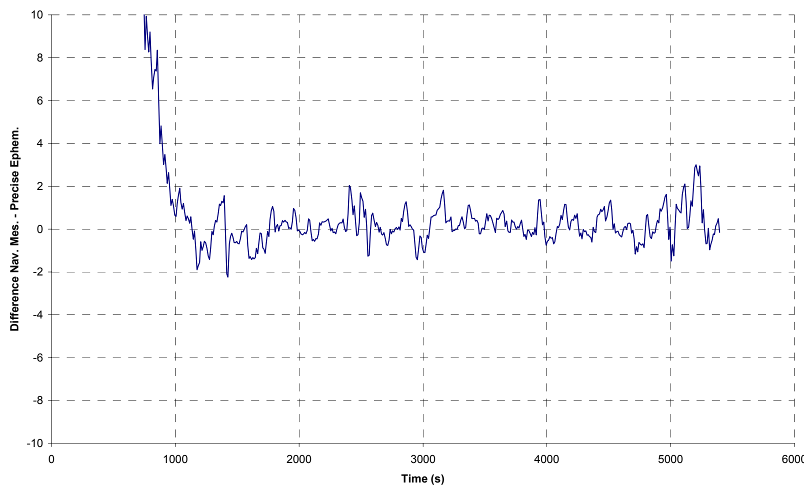

Figure 18 Difference in Position Module Error using Single Differences

Measurement gaps

Orbit determination evolution in relative navigation when a measurement gap appears should be also analyzed and in this subsection the behavior of the relative navigation strategies is shown.

The measurement gap has been introduced only in the measurement file corresponding to the chaser spacecraft.

Recovery performances in real-time navigation are shown in the following table.

Table 6 Recovery Performances in Relative Real-Time Navigation

Gap Duration (min)	Recovery Time (min)	K	Algorithm
5	2.5	0	SV
5	< 0.5	0	SD
10	8	3	SV

Gap Duration (min)	Recovery Time (min)	K	Algorithm
10	< 0.5	0	SD
20	No recovery	All	SV
20	4.5	6	SD

Where K represents the number of rejected measurements of the first available set of measurements after the measurement gap; SV indicates State Vector algorithm and SD Single Difference algorithm.

When the gap duration is set to 20 minutes there is no recovery if the state vector differences algorithm is used. The following sets of measurements corresponding to the chaser vehicle are rejected and then the recovery is not possible, the application of the rejection criterion should be reviewed.

The behavior of the single difference algorithm when there is a measurement gap is better than in the state vector difference algorithm.

CONCLUSIONS

The validity of the navigation message use for ATV absolute and relative navigation strategies has been proven and has answered to the real time data availability constraint.

Relative navigation strategies show similar results for both considered algorithms in the reduction of SA contributions. Due to spatial proximity between the ATV and ISS and same GPS antennae configuration, the selected GPS satellites for both spacecraft are the same. This means both spacecraft have similar SA errors and similar GPS satellite clock errors.

The GPS ephemeris errors due to consider navigation message information has been analyzed and the results are also similar due to the same above-mentioned reasons.

Finally the behavior in relative navigation when measurement gaps appears has been considered and the best results corresponds to the single differences algorithm.

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